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VTOL TRANSPORT AIRCRAFT

FC Comparative Study

PRELIMINARY WING WEI ON
VERTOL REPORT

57AA 42111

VERTOL Aircraft Corporation

formerly - Piasecki Helicopter Corporation

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Comparative Study of Various Types of VTOL Transport Aircraft

PRELIMINARY WING WEIGHT DETERMINATION
REPORT R-81

Vertol Aircraft Corporation Morton, Pennsylvania

ONR ARMY

A Combined

Research and Development Program

Contract NONR 1681(00)

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I. SUMMARY

The following conclusions may be reached from this investigation:

1. As shown in Figure I, the tilt wing design lends itself to more efficient wing structural design than does the vectored lift. It is, however, important to note that the weight differential is not large over the greater portion of range of parameters investigated. Therefore, the structural weight of the wing is not an important consideration in the choice between the tilt wing and vectored lift designs.

2. As shown in Figures II through IV, the tilt wing design indicates a wing weight to gross weight ratio between 4 1/2% and 16 1/2% at a taper ratio of .5, between 4 3/4% and 18% at a taper ratio of .75, and between 5% and 18% at taper ratio of 1.00, over the range of wing spans from 50 to 150 feet and aspect ratios from 5 to 12. The short span, low aspect ratio wings have the best weight ratios.

3. As shown in Figure V, the vectored lift design indicates a wing weight to gross weight ratio between 4% and 21% at a taper ratio of 1.0, which was the only taper ratio investigated. Wing span range was from 50 to 150 feet and aspect ratio range was from 5 to 12. The short span, low aspect ratio wing has the best weight ratio. It should be noted that above approximately 120 foot span the 7.65 aspect ratio wings have more favorable weight ratios than the 5.0 aspect ratio wings.

4. Weight estimation methods of Reference 1 show reasonable agreement with calculated values for tilt wing and vectored lift designs, agreeing within 10% for the two cases checked (Reference pages 13 and 14).

SYMBOLS

a	-	$\frac{dC_L}{d\alpha}$ per radian
A.R.	-	aspect ratio
b	-	span of wing
b_1	-	semi span of wing
c	-	chord
c_r	-	root chord
c_t	-	tip chord
C_L	-	section lift coefficient
$C_{l_{a1}}$	-	section lift coefficient on wing for an overall value of $C_L = 1.0$
C_{Mac}	-	moment coefficient about the aerodynamic center
f_b	-	bending stress
f_d	-	direct stress
F_b	-	allowable bending stress
F_d	-	allowable direct stress
g	-	acceleration of gravity = 32.2 ft./sec.^{2}
G.W.	-	gross weight
h_1	-	vertical distance between spar cap centroids
h_2	-	horizontal distance between spar cap centroids
L_a	-	dimensionless lift coefficient = $C_{l_{a1}} \frac{c.b}{S}$
η_z	-	vertical load factor (limit)
η_x	-	chordwise load factor (limit)
q	-	dynamic pressure in pounds per sq. ft.

S	-	wing area, sq. ft.
τ	-	Torque, inch lbs.
W	-	weight
W/S	-	wing loading, pounds per sq.ft.
X	-	fore and aft axis along fuselage
Y	-	lateral axis along wing
Z	-	vertical axis
λ	-	taper ratio (tip chord ÷ by root chord)

II. INTRODUCTION

Preliminary studies of Vertol Aircraft Corporation, (reference 3) indicate that two VTOL transport-type aircraft configurations require detailed wing weight studies to aid in determination of optimum VTOL types. These configurations are: (1) a tilting wing design, where wing and propellers rotate approximately 90° about a lateral tilt axis for vertical flight, and (2) a vectored lift design, where vertical flight is attained by deflecting the propeller slipstream downward, with a compound flap arrangement.

In final design, the main structural difference between these two types will be the wing configuration. It is important, therefore, to estimate a reasonably accurate wing weight, for a wide range of design parameters.

The parameters chosen and the range of each are shown in the Table below:

TABLE 1

PARAMETER	CONFIGURATION*	RANGE		
Gross Weight	(1) and (2)	60,000#	to	120,000#
Aspect Ratio	(1) and (2)	5.0	to	12.0
Taper Ratio	(1)	0.5	to	1.0
Taper Ratio	(2)	1.0		
Span Loading, Ww/b	(1) and (2)	800	to	1600

*(1) - Tilt Wing; (2)- Vectored Lift.

Note: For the vectored lift configuration, only a rectangular wing (taper ratio = 1.0) was considered, since a constant chord flap at a given percentage of the wing chord was assumed as a practical approach.

In estimating wing weight, the procedure used is to compute the amount of bending and shear material required to resist the loads given in the structural criteria. Simplifying assumptions were made to expedite the solutions. These are found on Page 8.

In order to investigate the large number of cases possible within the parameter ranges, the work was set up in tabular form for an IBM Digital Computer solution.

Wing weights for two examples were computed by the estimating method of reference (1) and compared with the IBM solutions. These comparisons are found on Pages 14 and 16.

III. DISCUSSION

Part I - Structural Criteria

A symmetrical loading for a 3.00 G limit maneuver condition was selected as a critical condition for machine solution. Take-off, landing, and one wheel landing conditions have been investigated manually and found to be less critical.

Spanwise airload distribution on the wings is done by the familiar Lotz method. Because only full span flaps are used on the vectored lift, the airload distribution is assumed the same for flaps down or flaps up.

A cross plot of airload parameters from Reference (2) was made for the flaps-up condition for various aspect ratios and tapers. Dimensionless airload parameters $L_a = C_{l_{a1}} \frac{c \cdot b}{S}$ were plotted against spanwise stations in percent of span, for aspect ratio values of 5, 7.65 and 12, and for taper ratios of 1.00, .75 and .50; see Page 4. This gave 9 sets of spanwise loadings in dimensionless form. The assumption was made that no aerodynamic twist is used, therefore $C_{l_b} = 0$. Each curve was graphically integrated to give a dimensionless airload shear and moment curve. The shear is then $C_L q S \int L_\alpha \, dx$, and the moment is $C_L q S \frac{b}{2} \iint L_\alpha \, dx\, dx$. The constants $C_L q S$ and $C_L q S \frac{b}{2}$ were combined with the dimensionless shear and moment values by the digital computer.

Spanwise variation of basic wing weight was estimated from conventional airplane wing design practice. Inertia loading due to

basic wing weight was plotted in a non-dimensional form against wing stations in percent of semi-span. This loading was graphically integrated twice to give non-dimensional shear and bending moments. Actual shear and bending moments were obtained by multiplying by design wing weight, and by design wing weight and semi-span respectively.

Chordwise loads include propeller thrust, chordwise air load components, and chordwise weight components.

The wing bending structure is assumed to be composed of 4 spar caps. No spanwise stiffeners are used; but ribs act as skin stiffeners. This structure is reasonable for a wing which resists high chordwise as well as vertical bending. Spar webs are designed by vertical shear and wing torsion. Wing skin around the leading edge resists torsional shear only, while skin aft of the leading edge resists torsion and chordwise shear. The torsion has been distributed to the wing torque boxes by unit torque factors evaluated by manual solution of torque distribution equations for specific geometric properties.

Total spar cap area is determined by (1) computing the area required to resist vertical bending by the formula $f_b = \frac{Mc}{I}$, or $\Sigma A h_1^2 = \frac{Mc}{F_b}$ and (2) obtaining the area required to resist chordwise bending by the formula $P = \frac{M}{h_2}$ and $f_d = \frac{P}{\Sigma A} = \frac{M}{\Sigma h_2 A}$. The summation of these areas is the required spar cap area. In combining these areas for the flight condition, it will be noted that chordwise bending is predominantly negative (wing leading edge in compression), and vertical bending is positive (upper wing surface in compression).

The upper forward spar cap, and lower rear spar cap loads are additive, whereas the load on the other two caps due to vertical bending is of opposite sign to that due to chordwise bending. However, it is possible to have vertical bending with very little chord bending in certain flight conditions, consequently the two latter caps are designed by vertical bending only. Thus, the chord bending affects overall cap area only in the two caps where it adds to the load due to vertical bending. In calculating required cap areas, effective skin and webs have been neglected.

Part II - Assumptions

A. Tilt-Wing Configuration

1. Although an unsymmetrical condition such as one-wheel landing will design the spar webs across the fuselage, the assumption that the fuselage provides a reaction to the wing loads at ℄ of aircraft provides enough conservatism to ignore unsymmetrical conditions.

2. Three structural torque boxes carry through the fuselage.

3. Torque distribution between the three boxes is assumed to be the same as calculated for the sample case, regardless of skin or web gages.

4. Four wing spar caps are used. An average compression or tension allowable of 40,000 psi is assumed to be used throughout the wing span. This low value for the allowables plus neglecting effective skin in tension or compression is assumed

to compensate for the fact that no allowance is made for a practical minimum area for the spar caps (i.e., where the bending moments are near zero, the assumption is made that required spar cap areas are near zero).

5. Minimum web gages or skin gages are assumed to be .032. Although it is possible that in an actual aircraft gages lighter than .032 might be used, this assumption was adopted to partially compensate for excluding the weight of skin and web splices.

6. The wing has a straight taper from station zero to the tip for both chord taper and thickness taper.

7. Torque due to aileron deflection at wing tip is small enough to be neglected since the skin in this area is designed by minimum thickness considerations noted in item 4 above.

8. Wing thickness is 15% chord at root, and 12% chord at tip.

9. Propellers are mounted on the wing at 40% and 80% of the semi-span. The concentrated weights at these stations were set-up in percentages of the gross weight and may be varied to account for engine weight.

10. The weight of the mechanism required for tilting the wing is included in the total wing weight. The mechanism is assumed to consist of two hydraulic actuators, acting in parallel and mounted within the fuselage.

An estimation of the required cylinder size, using the 100,000# gross weight design as a basis, was made by assuming a wing center of pressure at 25% of the chord. The tilt axis is at 50% of the chord, and an 18" moment arm between the tilt axis, and actuator link fitting is considered. From this data, it was estimated that an actuator plus fitting weight of 200# would be required for the 100,000# gross weight aircraft. It was further assumed that this weight varied linearly with gross weight.

B. Vectored Lift Configurations

1. Assumptions 1, 4, 5, 6, 8 and 9 of tilt wing configuration apply.

2. Two structural torque boxes are used and carried through the fuselage. Torque distribution between the boxes is the same as calculated for the sample case, regardless of skin or web gages.

3. The forward flap is designed for all torque due to C_{Mac}, and dead weight about the aerodynamic center. The reason for this was to arbitrarily give a weight allowance for hinge mechanisms and operating structure. No extra allowance is made for flap spars as area required will be small. The assumption is made that forward flap is supported in at least five places. A single actuator operates the flap, at the side of the fuselage.

4. The weight of the aft flap is calculated for an area of 30% of wing area and a gage of .032. This will give allowances

for ribs and honeycomb filler, as well as actuator fittings.

5. The design torque is based upon take-off with full flap deflection plus torque due to prop overhang with $\eta_z = 2.0$ for take-off. This is used in combination with normal and chordwise shears determined for a maneuver load factor = 3.0. This is done to eliminate the necessity for checking a separate landing condition, since flaps down pitching moment will largely design skins.

IV. CALCULATIONS AND RESULTS

The following pages contain sample calculations of the Englebry wing weight estimation method (reference 1) compared with the values calculated by the digital computer method.

Also included are the curves of wing weight/gross weight ratio vs. span for each gross weight.

WING WEIGHT ESTIMATION - TILT WING

Sample Calculation

C. R. Englebry Method (Ref. 1)

Given: Gross Weight - 80,000#
Aspect Ratio - 5.0
Wing Span - 100'
Taper Ratio - .5

Calculation of Wing Weight

$b = 100' = 1200$ in.

$b'_{\ell} = 600$ in.

$$\text{Equivalent Root Chord} = \frac{2 \times b}{AR\,(1+\lambda)} = \frac{2 \times 1200}{5\,(1+.5)}$$

$$C_R = 320 \text{ in.}$$

$$\lambda = \frac{C_T}{C_R}$$

$$C_T = \lambda C_R = .5 \times 320 = 160 \text{ in.}$$

$$S = \text{Wing Area (in.)}^2 = \frac{C_R + C_T}{2} \times b = \frac{320 + 160}{2} \times 1200$$

$$S = 288{,}000 \text{ (in.)}^2$$

$$\text{T.F.} = \text{Taper Factor} = \left[\frac{1+\lambda}{6}\right] \cdot \left[\frac{.51\,K_r + \lambda^2 K_t}{K_r^2 + K_r K_t \lambda}\right]$$

$K_r = .15$ = % thickness at root

$K_t = .12$ = % thickness at tip

$$\text{T.F.} = \left[\frac{1+.5}{6}\right] \cdot \left[\frac{.51 \times .15 + .5^2 \times .12}{.15^2 + .15 \times .12 \times .5}\right]$$

T.F. = .845 (Estimating T. F. from T. F. Curves gives T. F. = .847)

L. F. = Load Factor = 3.0 (limit) = 4.5 (ultimate)

lw = Wing Loading (psi) $= \frac{G.W.}{S} = \frac{80,000}{288,000}$

lw = .278 psi

f = Average Allowable = 35,000 psi

C_1 = Non-Bending Material Factor
for Gross Weights between 40,000# & 140,000#.

$= .024 - .001 \frac{G.W. - 40,000}{100,000}$

$= .024 - .001 \left(\frac{80,000 - 40,000}{100,000}\right) = .024 - .001\ (.4)$

= .0236

$Ww = \text{Wing Weight} = C_1\, lw^{1/4} S + \frac{L.F. \times b_1^3 \times l_w \times T.F.}{2.5\, f}$

= 4940 + 2700

Ww = 7640#

Ww = 8403# by the IBM Digital Computer

Englebry Method Results & Digital Computer Results Compare within 10 percent.

WING WEIGHT ESTIMATION - VECTORED LIFT

Sample Calculation

C. R. Englebry Method

Given: Gross Weight = 80,000
Aspect Ratio = 5.0
Wing Span = 100'
Taper Ratio = 1.0

$$b' = \frac{b}{2} = \frac{100 \times 12}{2} = 600 \text{ in.}$$

$$CR = \frac{2 \times b}{AR\ (1 + \lambda)} = \frac{2400}{5 \times 2} = 240 \text{ in.}$$

$$CT = \lambda C_R = 1 \times 240 = 240 \text{ in.}$$

$$S = b \times C = 1200 \times 240 = 288{,}000\ (\text{in.})^2$$

$$T.F. = \left[\frac{1+\lambda}{6}\right]\left[\frac{.51 K_r + \lambda^2 K_t}{K_r^2 + K_r K_t \lambda}\right]$$

$$Kr = .15 \qquad Kt = .15$$

$$T.F. = \frac{2}{6}\left[\frac{.51 \times .15 + 1^2 \times .15}{.15^2 + .15 \times .15 \times 1}\right]$$

$$= \frac{1}{3}\left[\frac{.2265}{.0450}\right] = 1.68$$

$$lw = \frac{G.W.}{S} = \frac{80{,}000}{288{,}000} = .278$$

f = Average Allowable Stress = 35,000 psi

C_1 = .0236

L.F. = 3.0 (limit)
= 4.5 (ultimate)

$$Ww = C_1 \, lw^{1/4} S + \frac{L.F. \; b_1^3 \; lw \; T.F.}{2.5 \, f}$$

$$= \left[.0236 \times .278^{1/4} \times 288{,}000 \right] + \left[\frac{4.5 \times 600^3 \times .278 \times 1.68}{2.5 \times 35{,}000} \right]$$

$$= 4940 + 5200$$

Ww = 10,140#

Ww = 10,510# by the IBM Digital Computor

Englebry Method Results & Digital Computor Results Compare Within 4 percent.

FIGURE 1

ENVELOPE CURVES FOR VECTORED LIFT & TILT WING

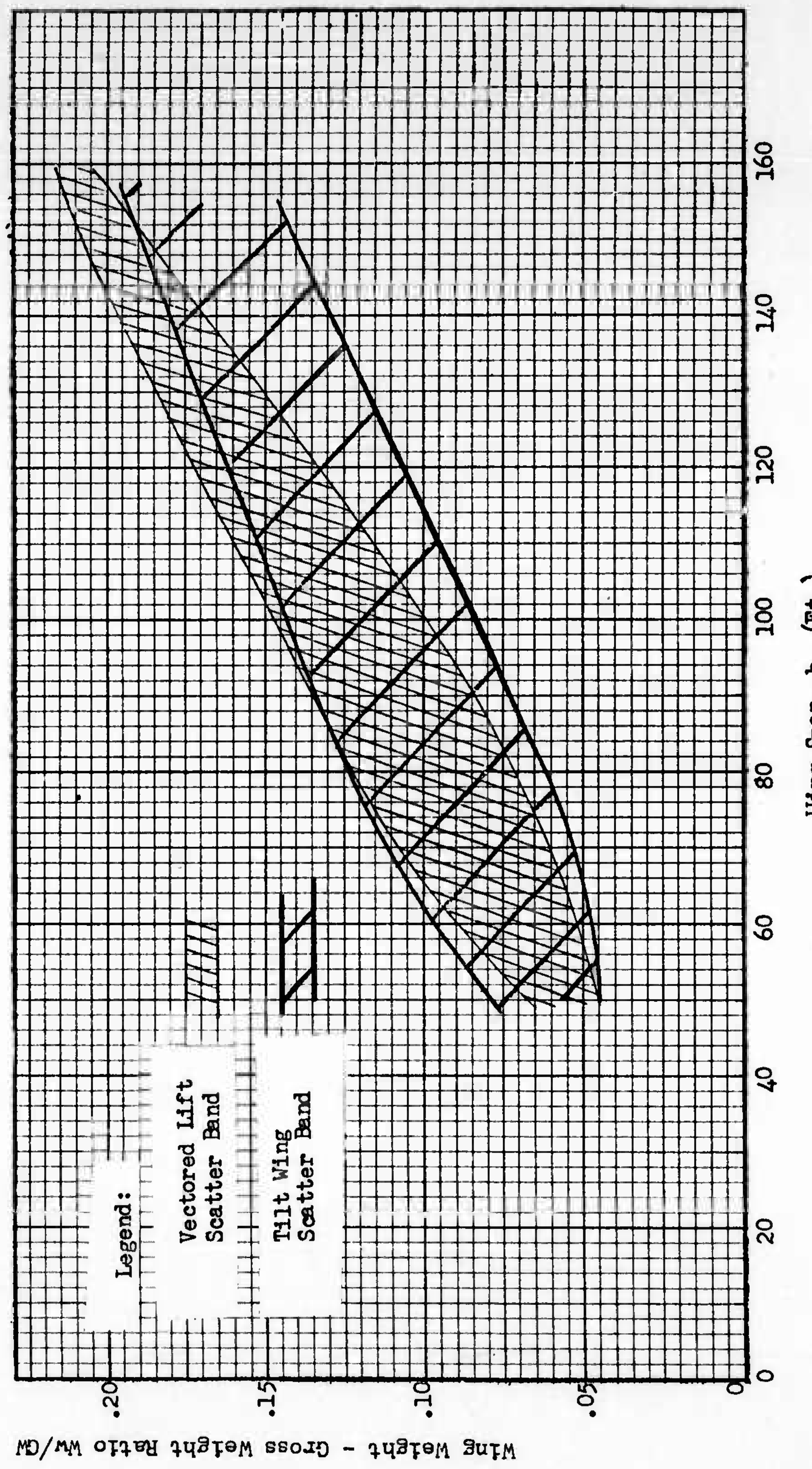


FIGURE 2

WING - GROSS WEIGHT RATIO VS WING SPAN FOR
SEVERAL ASPECT RATIOS. CHORD TAPER RATIO = .5

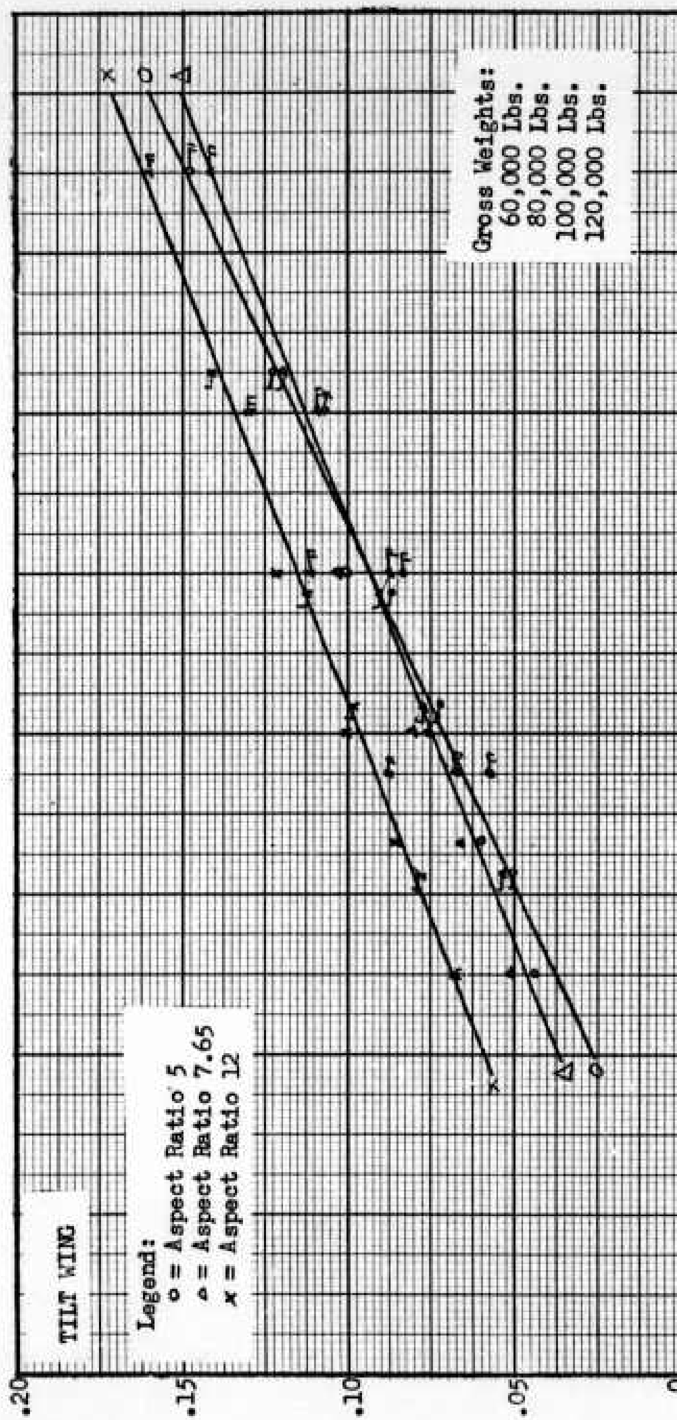


FIGURE 3

WING - GROSS WEIGHT RATIO VS WING SPAN FOR SEVERAL ASPECT RATIOS. CHORD TAPER RATIO = .75

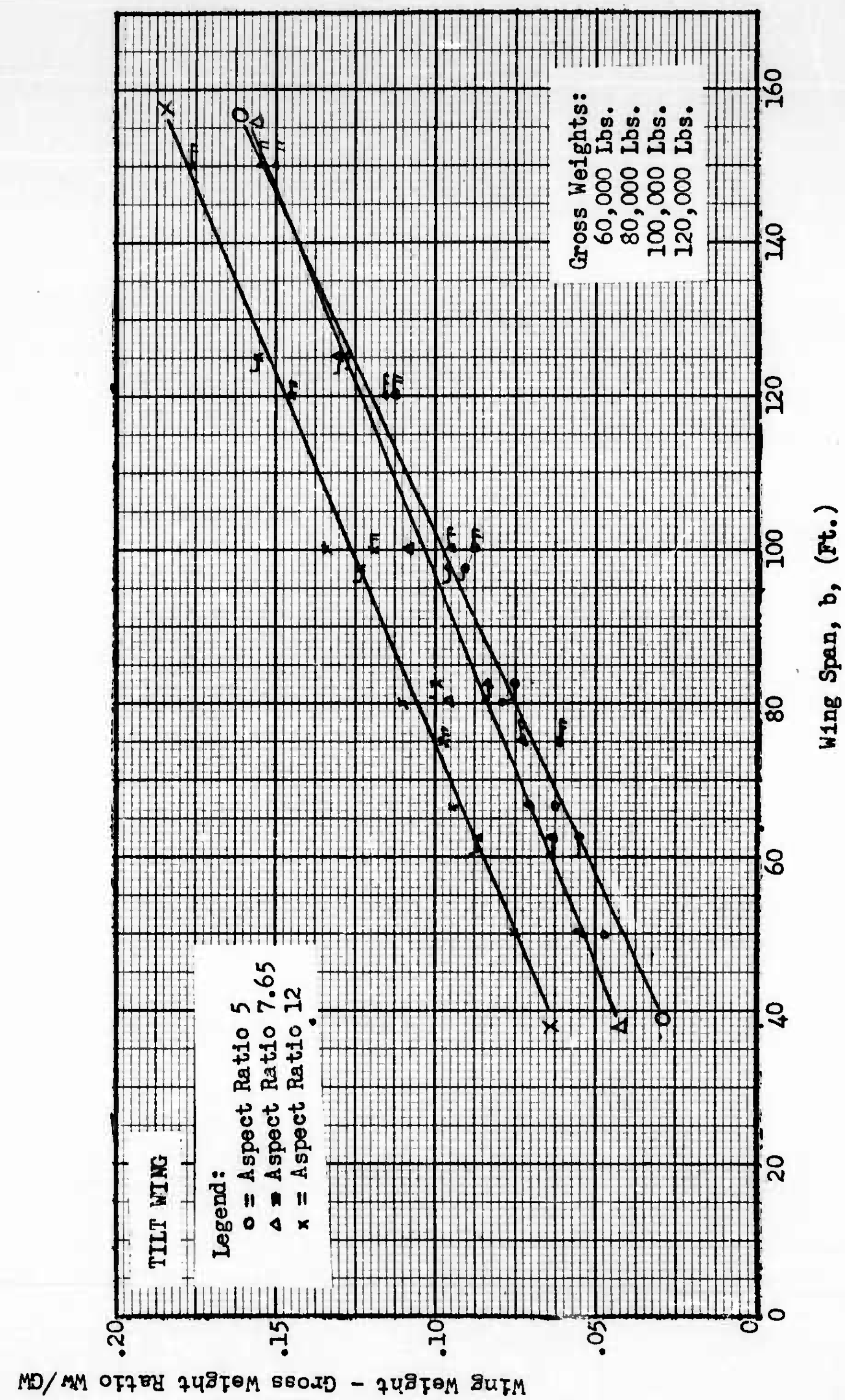


FIGURE 4

WING - GROSS WEIGHT RATIO VS WING SPAN FOR
SEVERAL ASPECT RATIOS.CHORD TAPER RATIO = 1

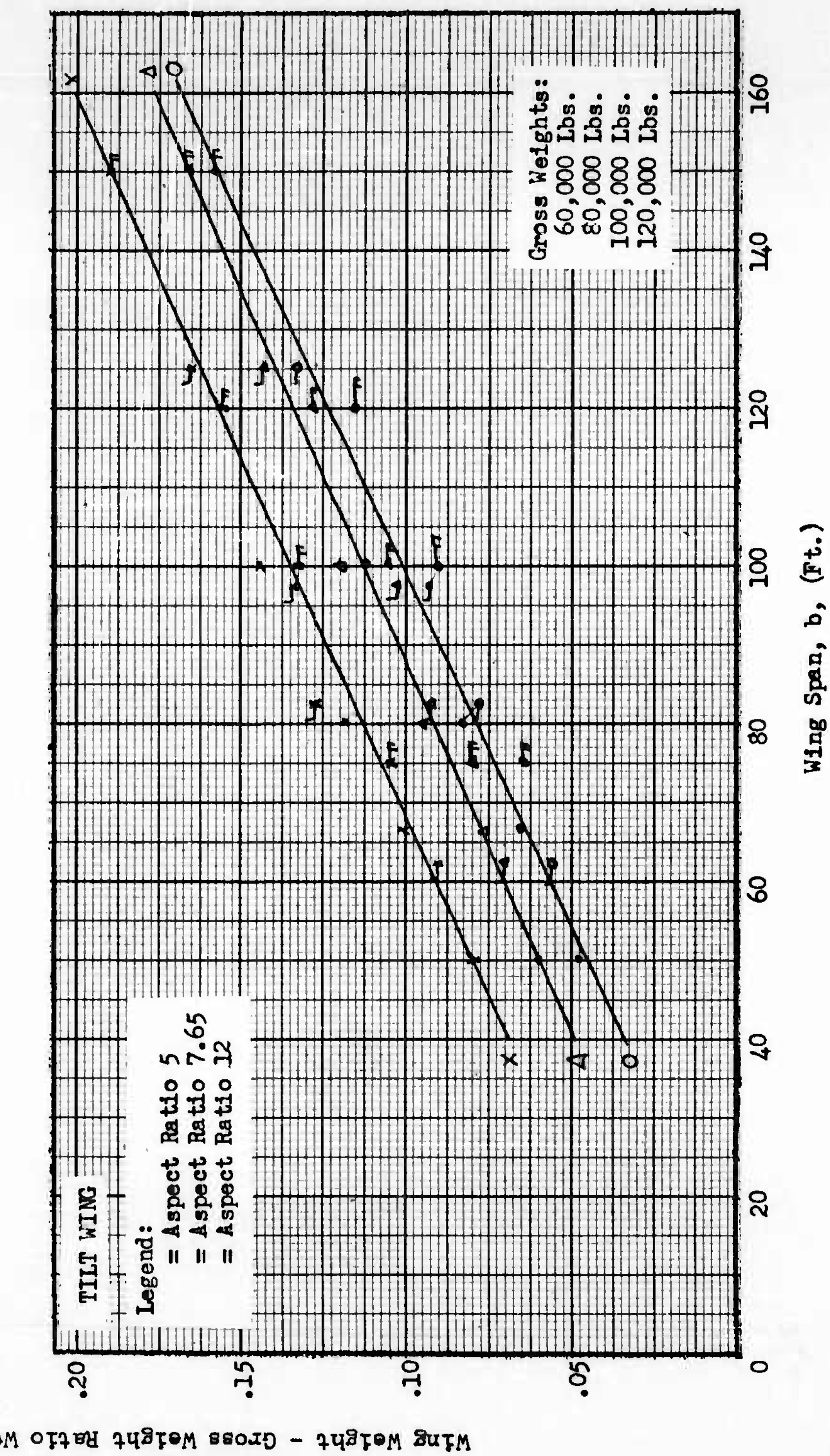
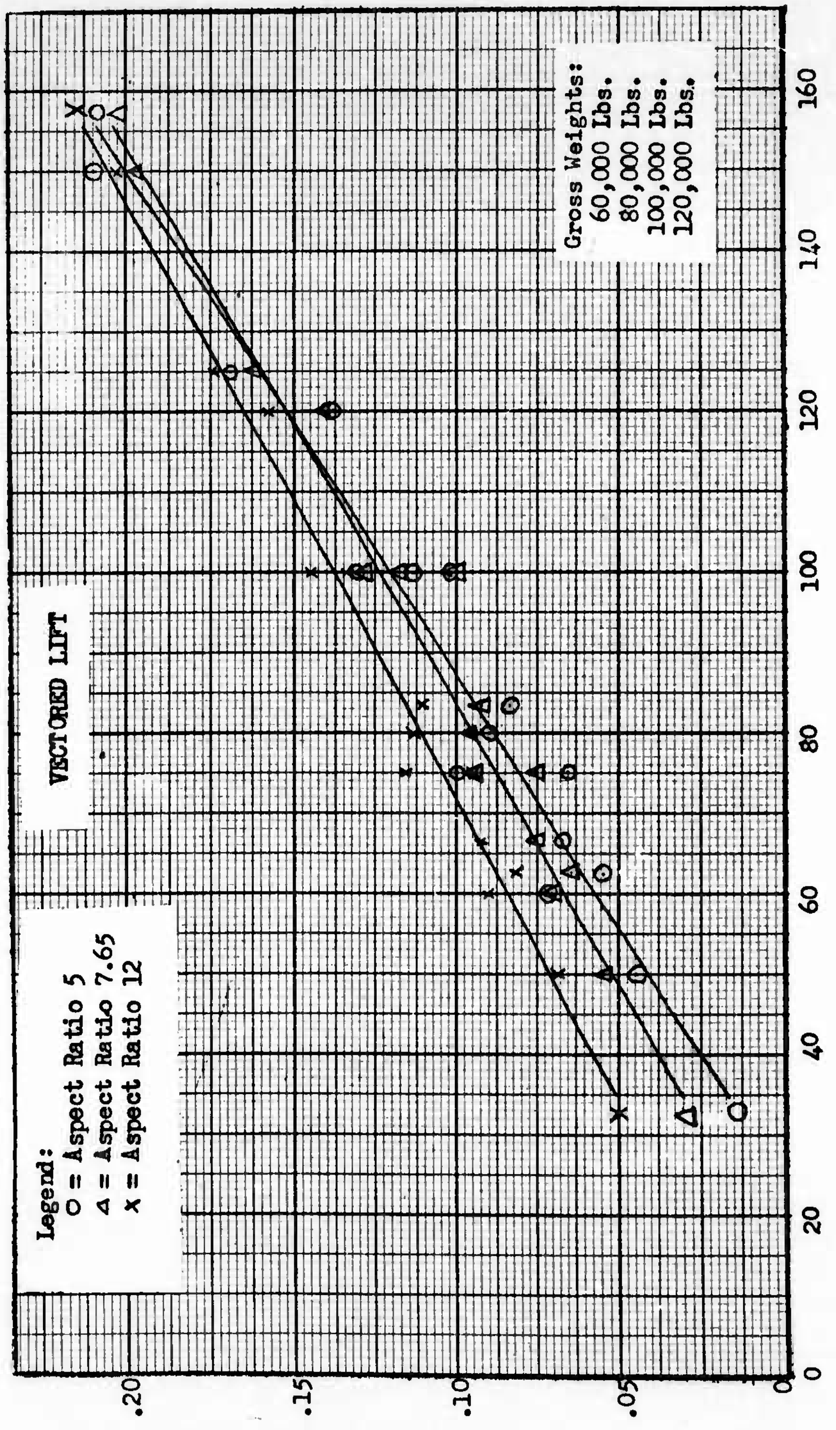


FIGURE 5

WING WEIGHT - GROSS WEIGHT RATIO VS WING SPAN
FOR SEVERAL ASPECT RATIOS. CHORD TAPER RATIO = 1.0



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